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March 30, 1972
Attachment B

Atmosphere Explorer (AE)
Spacecraft System Description

Prepared for:
Goddard Space Flight Center
National Aeronautics and Space Administration
Washington, D.C.
Under Contract No. NAS5-11432

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RCA | Government and Commercial Systems
Astro Electronics Division | Princeton, New Jersey

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PREFACE

This document provides a summary of the principal design and performance characteristics of the AE spacecraft system designed to support the Atmosphere Explorer C, D, and E missions. It has been prepared for the information of experimenters and other participants in the Atmosphere Explorer program as a general guide for design and operational planning. The description given herein represents the spacecraft system as defined at the conclusion of the Interface Definition Study.

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1.0 SYSTEM

1.1 General Description

In general configuration, the Atmosphere Explorer spacecraft is a 16 sided polyhedron, 53.5 inches in outside diameter and 45 inches high weighing 1282 lbs. The spacecraft contains a 3 axis attitude control system utilizing a momentum wheel to provide roll-yaw stiffening and pitch orientation, and magnetic torque coils to maintain momentum axis orientation in inertial space. A thruster and monopropellant hydrazine fuel supply is used to provide orbital apogee and perigee adjust capability throughout the 1 year life of the spacecraft. An active thermal control system maintains spacecraft temperature within operating limits. Command and communication systems are compatible with the consolidated MSFN/STADAN network. Power is obtained from a skin mounted solar cell array. The spacecraft is designed to be launched by a Delta vehicle into near-polar, near-equatorial or medium inclination orbits, with nominal apogee at 4000 km, and perigee at 150 km. The spacecraft is designed to support the experiment complements identified for the AE-C, D and E missions, the most complex, in terms of the number of experiments accommodated, being the AE-C mission. The AE-C complement includes 14* experiments of 184 lbs total weight requiring 115 watts of regulated power. The spacecraft block diagram and configuration are shown in Figure 1 and Figure 4.

1.2 Spacecraft Coordinate System

The spacecraft coordinate system is a right hand system as shown in Figure 2. The origin is taken at the geometric center of the spacecraft, on the axial centerline midway between the end surfaces. The coordinate system is body fixed; the orbital directions shown are for reference only.

1.3 Nomenclature

1.3.1 Definitions

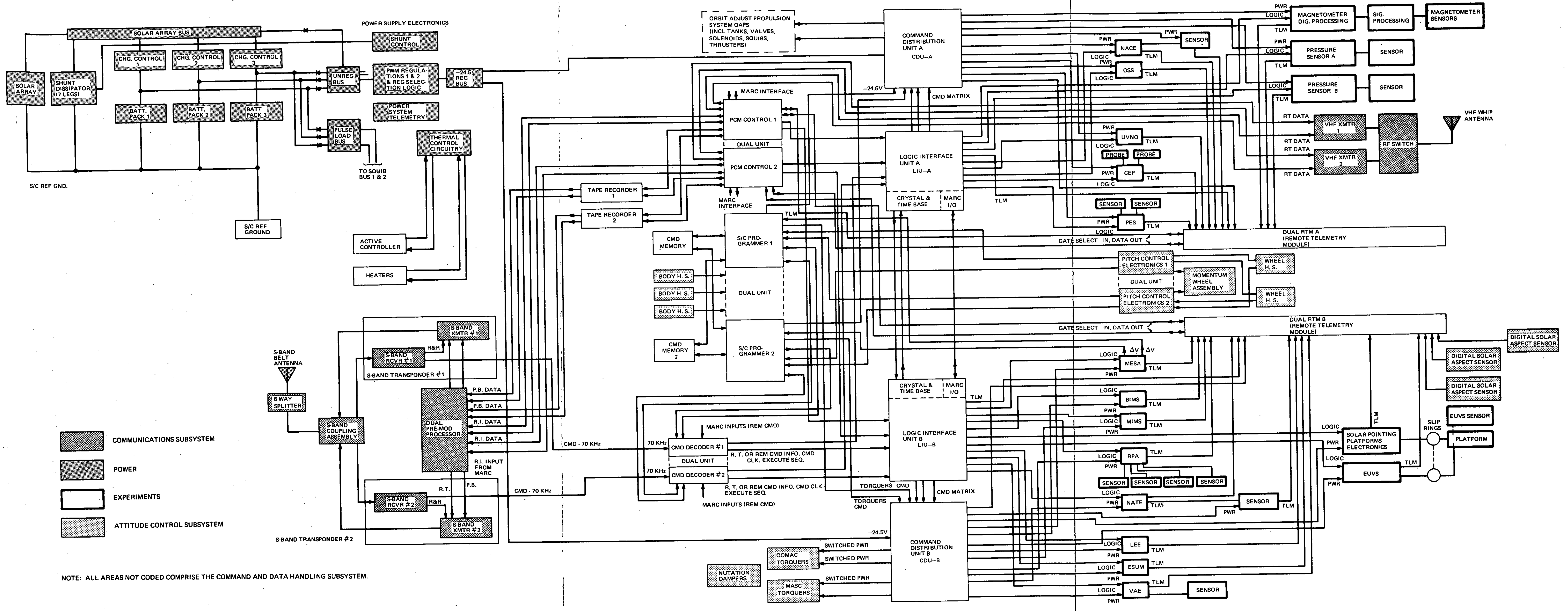
- spacecraft top -Z axis
- spacecraft bottom +Z axis
- sun angle - angle between +Z axis and sun line

*A fifteenth (magnetometer) experiment is presently under consideration.

FOLDOUT FRAME 1

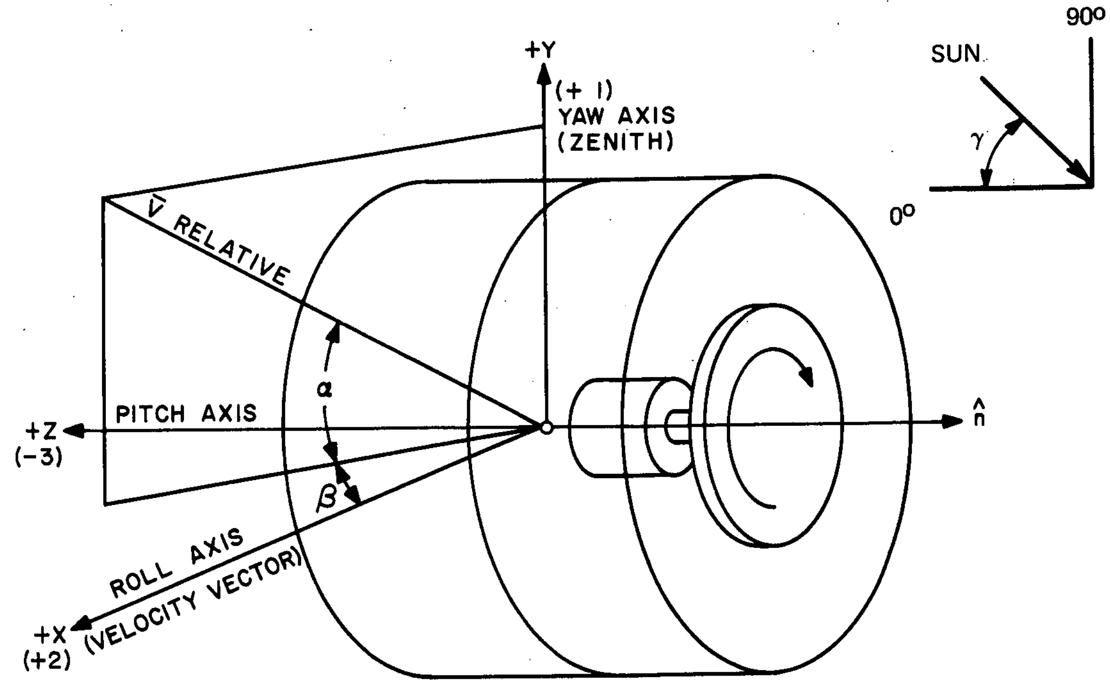
FOLDOUT FRAME 2

FOLDOUT FRAME 3



NOTE: ALL AREAS NOT CODED COMPRISE THE COMMAND AND DATA HANDLING SUBSYSTEM.

Figure 1. Spacecraft System, Functional Block Diagram



Item	Axis System	Numerical Axis
Roll	X	2
Yaw	Y	1
Pitch	Z	-3
Wheel Rotation	Right Hand (CCW) About +Z	Right Hand About -3
Velocity Vector	Along +X	Along +2
Zenith	Along +Y*	Along +1
Orbit Positive Normal	Along -Z*	Along +3

*Initial mode - prior to 180° yaw maneuver required during C&E missions to maintain $\gamma \leq 90°$.

α - Angle of Attack

β - Angle of Sideslip

Figure 2. Spacecraft Coordinate System

- pitch -Z axis
- roll - +X axis
- yaw - +Y axis

1.3.2 Spacecraft Hardware

The family tree of spacecraft subsystems and components is shown in Figure 3. A list of Standard abbreviations used for the spacecraft components where required are given below.

Item	Abbreviation
Structural Subsystem	
Baseplate	BP
Central Column	CC
Separation Adapter	SR
Attitude Control Subsystem	
Momentum Wheel Assembly	MWA
Pitch Control Electronics A	PCE A
Pitch Control Electronics B	PCE B
Nutation Damper	ND
Attitude Control Torquer	ACC
Spin Control Torquer	SCC
Digital Solar Aspect Indicator	DSI
Solar Gate Sensor	SGS
Magnetometer	MGT

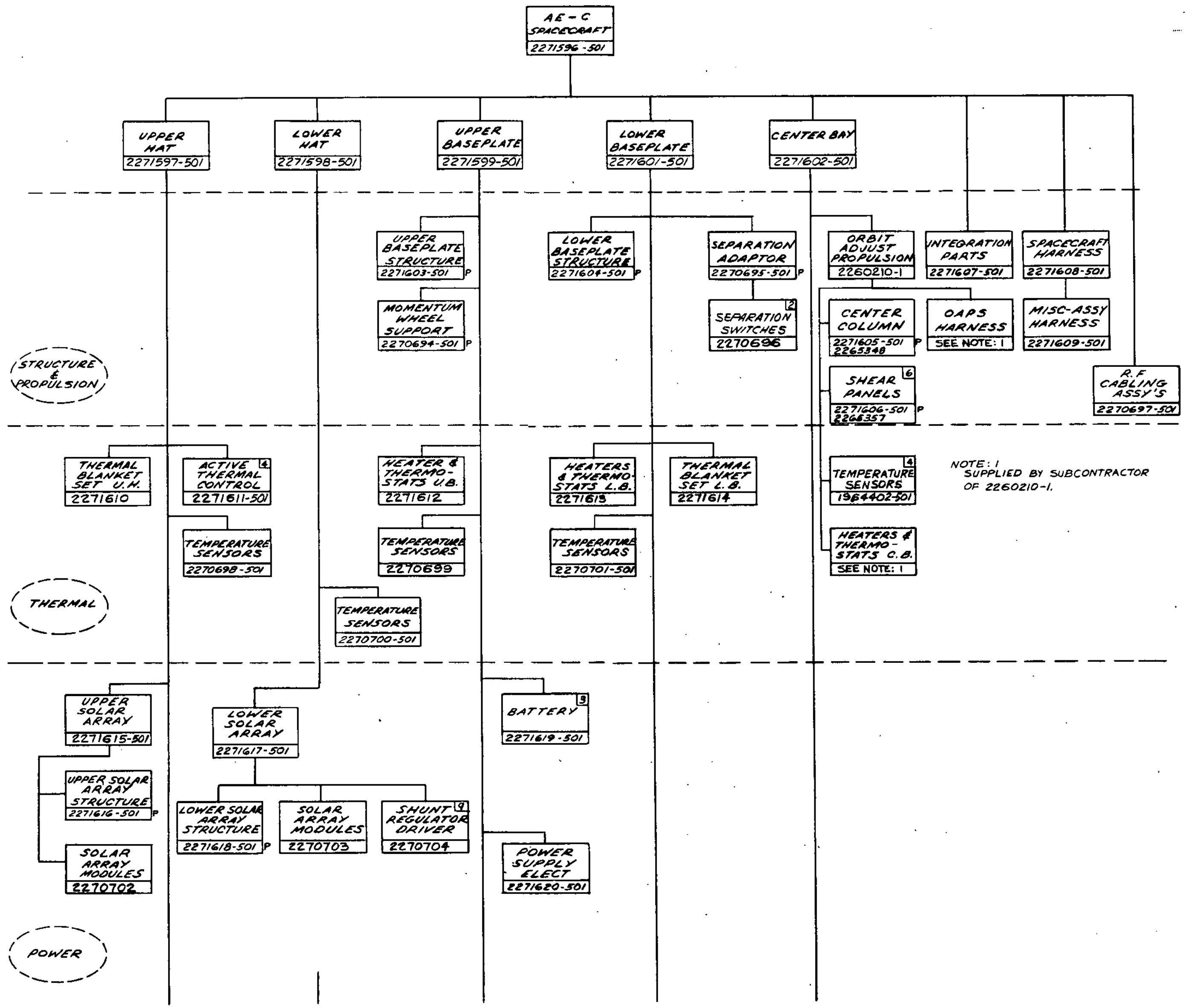


Figure 3. Atmosphere Explorer Missions C, D, and E, Family Tree (Sheet 1 of 4)

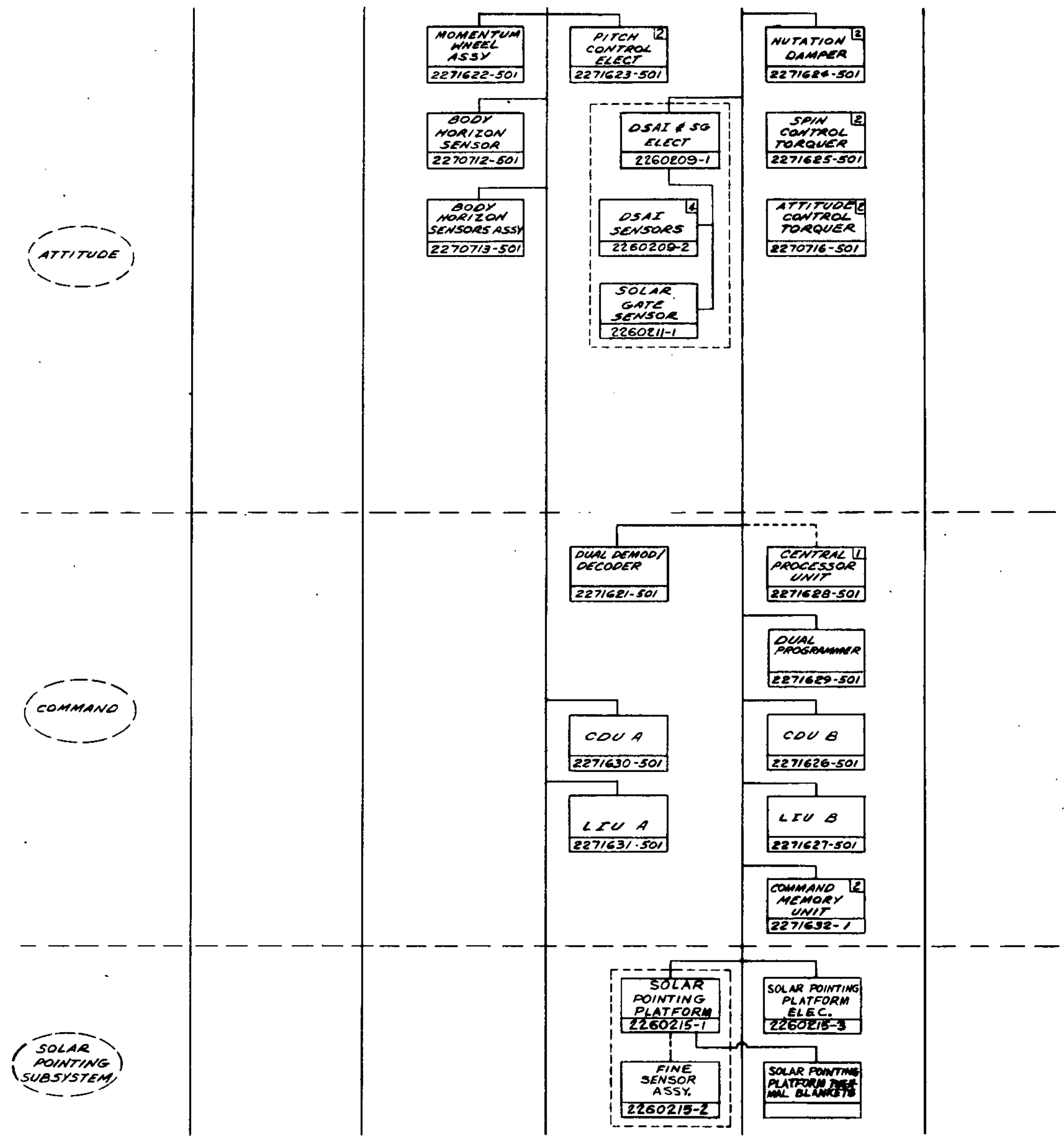


Figure 3. Atmosphere Explorer Missions C, D, and E, Family Tree (Sheet 2 of 4)

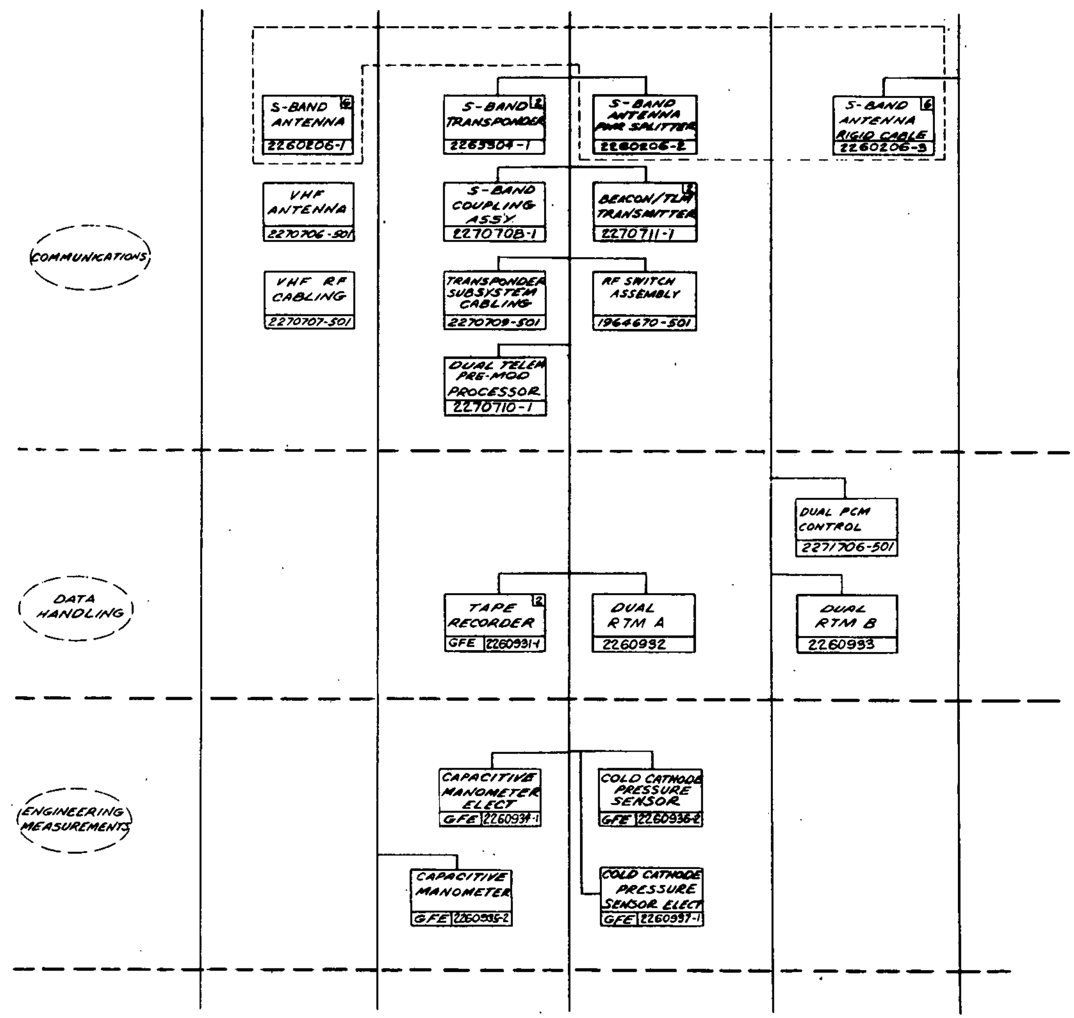


Figure 3. Atmosphere Explorer Missions C, D, and E, Family Tree (Sheet 3 of 4)

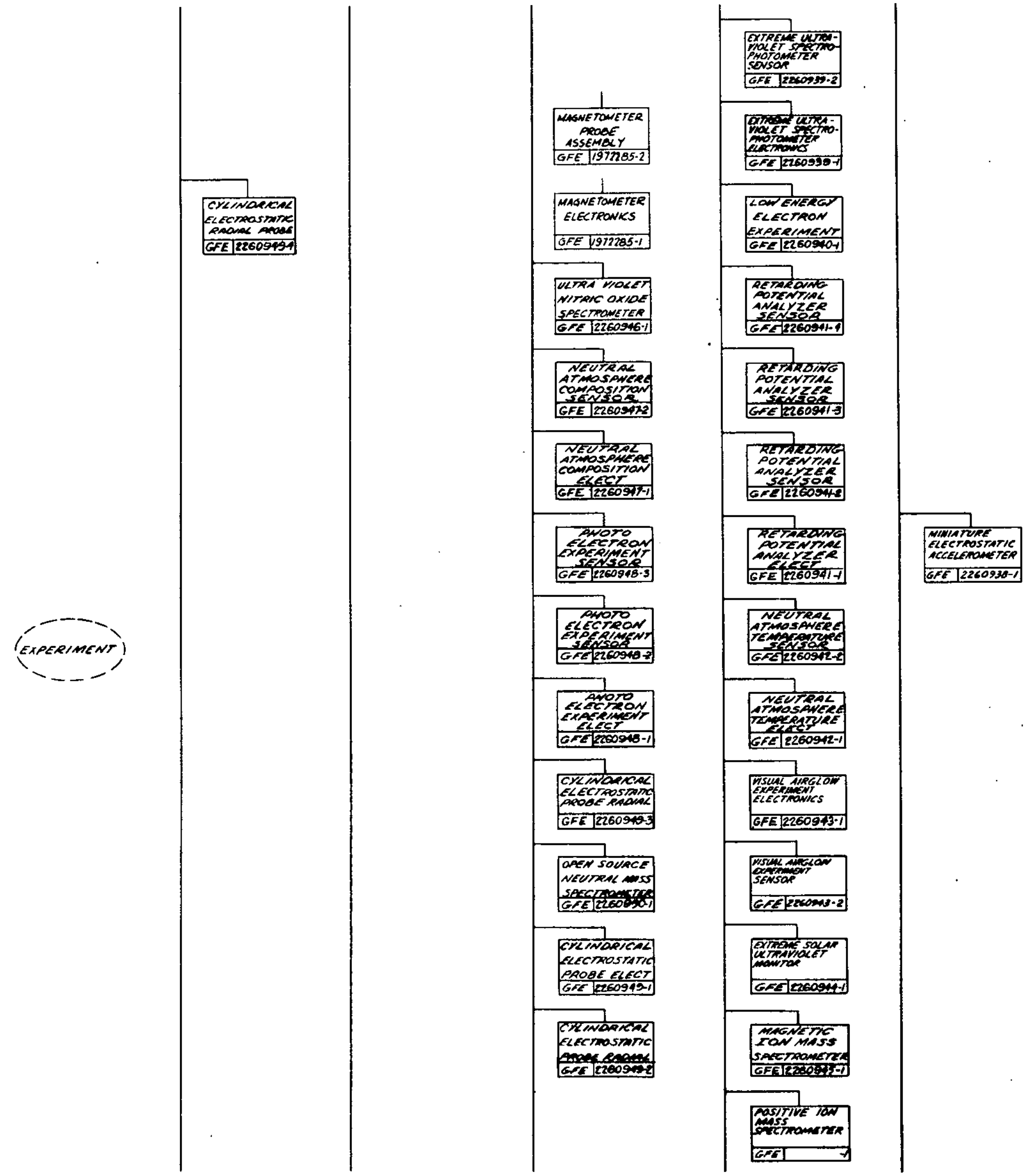


Figure 3. Atmosphere Explorer Missions C, D, and E, Family Tree (Sheet 4 of 4)

1.3.2 Spacecraft Hardware (Cont'd)

Item	Abbreviation
Body Horizon Sensors	BHS
Orbit Adjust Propulsion Subsystem	
Propellant Tanks	PPT
Rocket Engine Assembly	REA
Power Subsystem	
Upper Solar Array	SAU
Lower Solar Array	SAL
Power Supply Electronics	PSE
Shunt Dissipator	SD
Battery A	BPA
Battery B	BPB
Battery C	BPC
Communications	
VHF Beacon/Telemetry Transmitter	BTT
Antenna Network	AN
VHF Antenna	VHA
Range & Range Rate Transponder	RRR
S-Band Antenna	SBA
Pre-Modulation Processor	PMP

1.3.2 Spacecraft Hardware (Cont'd)

Item	Abbreviation
Data Handling Subsystem	
PCM Controller	PCM C
Remote Telemetry Module A	RTM A
Remote Telemetry Module B	RTM B
Tape Recorder 1	TR 1
Tape Recorder 2	TR 2
Command and Control Subsystem	
Dual Decoder	DD
Dual Programmer	DP
Memories	MM
Command Distribution Unit A	CDU A
Command Distribution Unit B	CDU B
Logic Interface Unit A	LIU A
Logic Interface Unit B	LIU B
Engineering Measurements Subsystem	
Pressure Sensor A	PSA
Pressure Sensor A Electronics	PSAE
Pressure Sensor B	PSB

1.3.2 Spacecraft Hardware (Cont'd)

Item	Abbreviation
Solar Pointing Subsystem	
Solar Pointing Platform	SPP
Solar Pointing Electronics	SPE
Miscellaneous	
Wire Harness	WH
Hardware	HW
Balance Weights	BW
Experiments	
Cylindrical Electrostatic Probe (Brace)	CEPE
Positive Ion Mass Spectrometer (Brinton)	BIMS
Photoelectron Spectrometer (Doering)	PES
Planar Ion Trap (Hanson)	RPA
Magnetic Ion Mass Spectrometer (J. Hoffman)	MIMS
Low Energy Electron (R. Hoffman)	LEE
Atmospheric Density Accelerometer (Champion)	MESA
Open Source Neutral Mass Spectrometer (Nier)	OSS
Neutral Atmosphere Composition (Pelz)	NACE
Neutral Atmosphere Temperature (Spencer)	NATE

1.3.2 Spacecraft Hardware (Cont'd)

Item	Abbreviation
Ultra Violet Nitric Oxide (Barth)	UVNO
Airglow Photometer (Hays)	VAE
Solar UV Filter Photometer (Heath)	ESUM
Solar UV Spectrophotometer (Hinteregger)	EUVS

1.4 Spacecraft Summary

1.4.1 Performance Summary

A summary of spacecraft performance is given in Tables 1 and 2.

1.4.2 Operational Features

Some of the principal operational features of the spacecraft are shown in Table 3.

1.4.3 Spacecraft Weight

The spacecraft weight list is shown in Table 4.

1.5 Configuration and Structure

1.5.1 Configuration

1.5.1.1 External Configuration

The spacecraft configuration is shown in Figure 4. The configuration is that of a 16 sided polyhedron, 53.5 inches in outside diameter and approximately 45 inches high. An extended flat is incorporated on the +X surface of the spacecraft to provide an unobstructed 2 π field of view for the forward facing experiments. A separation ring, which forms the mounting interface between the booster and the spacecraft, is mounted to the central column on the +Z end. A four-element VHF turnstile antenna array* is mounted to the -Z end of the

*An alternative simple dipole whip antenna is currently under study.

TABLE 1. SYSTEM PERFORMANCE SUMMARY

Life Time	>12 months
Projected Area	2400 in^2
Weight	1282 lbs
Thermal Performance	
Upper Baseplate	0°C to 20°C
Lower Baseplate	5°C to 35°C
Attitude Control Performance	
Roll/Yaw	<2° rms @ 120 km (5° max during orbit adjust operations)
Pitch (despun)	<2° rms
Pitch (spinning)	±1% of commanded rate
Nadir Pulse	±2°
Propulsion	
Propellant Capacity	370 lbs
Deliverable Impulse	75000 lbs sec (min)
ΔV per burn capability	24 ft/sec (within ±5° roll limit)
Command and Data Handling	
Memory Capacity	2 x 32 kilobits
Major Mode Comands Available	496
Record Capability	120 minutes per recorder
Experiment Power	
BOL	115 watts @ 20%
8 month-near polar	115 watts @ 15.5%
Experiments	
Weight	184 lbs
Power	115 watts
Mounting Area	1335 sq inches

TABLE 2. LOW PERIGEE PERFORMANCE

Item	Baseline Performance
Roll Angle Error at 120 km	
First Perigee Peak Error	2.3 degrees
Continuous Contacts	1.5 degrees
Momentum Change/Orbit	1.4%
Lost Contact Performance	
Minimum Component Temperature	-5°C
No Contact after Orbit Adjust, ± 3-degree Roll Limit (Automatic Roll Control)	Maintained for 12 orbits
Thermal Altitude Limits*	
Spinning Perigee in Sun	114 km (100% margin at 120 km)
Despun Perigee in Sun	129 km
Despun Perigee in Eclipse	121 km

*Spacecraft capability. Experiment thermal considerations may limit lowest perigee.

TABLE 3. OPERATIONAL FEATURES

Experiment Programming	Up to 72-hours delay time with 4 sec granularity. On-time selectable to 4 second granularity.
	Pulse or data-loading commands
Science and Status Telemetry Readout	S-Band/VHF/real time 16384 bits/sec. S-Band Playback 131072 bits/sec.
Command	S-Band 1024 bits/sec.
Orbit Determination	S-Band Range and Range Rate Tracking
Orbit Adjustment	Up to 24 ft/sec with < 1 ft/sec granularity (thrust range 4.1 lbf to 0.75 lbf)
Attitude Control	Mode change (despin/spinning) by direct momentum transfer. Roll/yaw control by magnetic torquing.
Attitude Determination	Horizon/Solar Sensor data: ground processing.

TABLE 4. BASELINE SYSTEM WEIGHT SUMMARY

ITEM	Weight (lbs)	
Structure Subsystem		
Baseplates and Brackets	54.4	
Central Column and Adapters	20.0	
Upper Array Substrate	17.7	
Lower Array Substrate	17.7	
		110.0
Attitude Control Subsystem		
Momentum Wheel Assembly	48.0	
Pitch Control Electronics A	7.0	
Pitch Control Electronics B	7.0	
Nutation Damper	4.0	
Attitude Torquers	6.4	
Momentum Torquers	1.6	
DSAI Electronics and Sensors	5.0	
Solar Gate Sensor	0.1	
Magnetometer	1.5	
Body Horizon Sensors	3.0	
		84.0
Data Handling Subsystem		
Dual PCM Controller	7.0	
RTM A	6.0	
RTM B	6.0	
Tape Recorders (2)	26.0	
		45.0
Command and Control Subsystem		
Dual Programmer	8.0	
Memory (2)	14.0	
Logic Interface Unit A	10.0	
Logic Interface Unit B	9.0	
Command Distribution Unit A	4.0	
Command Distribution Unit B	4.0	
Dual Demodulator/Decoder	4.0	
		53.0
Orbit Adjust Propulsion Subsystem		
Propellant Tanks	105.0	
Thrusters	1.2	
Valves, Piping and Misc.	26.0	
		132.2

TABLE 4. BASELINE SYSTEM WEIGHT SUMMARY (Continued)

ITEM	Weight (lbs)	
Power Subsystem		
Solar Array (exclusive of substrate)	46.0	
Power Supply Electronics	13.0	
Shunt Dissipators	3.0	
Batteries (3)	60.0	
		122.0
Communications and Ranging		
VHF Antenna	2.4	
VHF Antenna Network	2.2	
Beacon/Telemetry Transmitter	3.6	
Range/Rate Transponder (2)	20.0	
S-Band Coupling Assembly	4.0	
Dual Pre-Modulation Processor	5.0	
S-Band Antenna	5.0	
S-Band Power Splitter	1.0	
VHF RF Switch	0.4	
		43.6
Engineering Measurement Subsystem		
Cold Cathode Pressure Sensor	2.5	
Cold Cathode Pressure Sensor Elect	3.5	
Capacitance Nanometer	3.0	
Capacitance Nanometer Elect	3.2	
		12.2
Solar Pointing Subsystem		
Solar Pointing Platform	25.0	
Solar Pointing Electronics	6.0	
		31.0
Support Equipment		
Harness	40.0	
Hardware	10.0	
Balance Weights	20.0	
Baseplate Heaters	6.0	
Thermal Blankets	10.0	
Active Thermal Controllers	5.0	
Temperature Sensors	0.5	
		91.5

TABLE 4. BASELINE SYSTEM WEIGHT SUMMARY (Continued)

ITEM	Weight (lbs)	
Total Spacecraft	724.5	
Total Experiments	184.0	
Propellant	370.0	
Pressurant	3.4	
Total		1282.0
Experiments		
CEP	4.0	
BIMS	7.6	
PES	10.0	
RPA	11.0	
MIMS	10.0	
LEE	9.5	
MESA	16.0	
OSS	15.0	
NACE	16.0	
NATE	15.2	
UVNO	15.5	
VAE	12.0	
ESUM	18.1	
EUVS	24.0	
Total		184.0

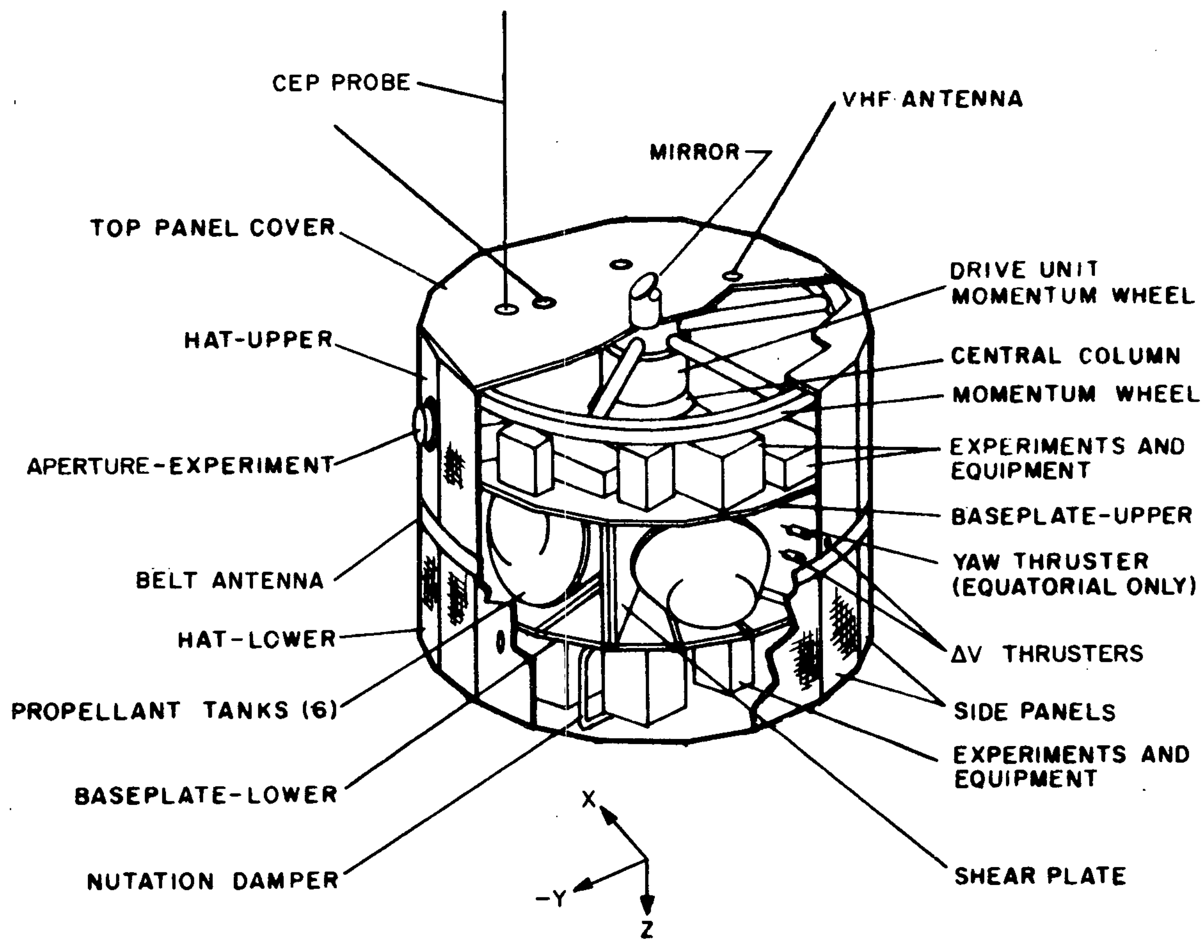


Figure 4. Spacecraft Configuration

spacecraft. An omni-directional S-Band belt antenna is mounted on the spacecraft equator. Spacecraft surfaces, except for experiment viewing ports, are covered with solar cells with cover glass, or are bare conductive surfaces to provide for spacecraft field grounding. Two electron temperature probes are mounted externally, one parallel to the spacecraft spin axis on the -Z surface and one mounted parallel to the spacecraft Y axis near the spacecraft equator. The spacecraft spin axis is oriented nominally perpendicular to the orbit plane during orbital operations (normal to the relative air-stream during low perigee operation.) Experiment sensors view the external environment through apertures provided in the solar array. Two thrusters utilized during orbit adjust maneuvers are oriented along the orbital velocity vector (spacecraft X axis) with the thrust line passing through the spacecraft C.M. An additional (canted) thruster is included on the 'E' mission to accomplish 180° yaw maneuvers. Approximately 1200 square inches of bare area is provided on the spacecraft outer surfaces to provide plasma grounding.

1.5.1.2 Internal Configuration

The internal configuration consists of two baseplates separated by a central column and connected by six shear ties. Spacecraft components and experiments are mounted to one side of each baseplate. Six cono-spherical shaped propellant tanks, carrying 370 lbs of hydrazine propellant, are grouped symmetrically about the central column, sandwiched between the two baseplates. The momentum wheel assembly for spacecraft stabilization is mounted on one end of the central column within the spacecraft, with its attitude sensing mirror assembly projecting through a hole in the center of the upper surface. The solar pointing platform is contained within the bottom end of the center column adjacent to the booster interface.

1.5.1.3 Experiment Mounting

The majority of the experiments are mounted along the outer edge of the base plates, oriented to view the velocity vector, the zenith, or the general environment. The sun pointing Extreme Ultra-Violet Spectrometer is mounted within the +Z end of the central column, near the +Z end of the solar array. The Atmospheric Density Accelerometer is mounted within the central column on the spin axis. The baseplate layouts for the AE-C mission are shown in Figure 5.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

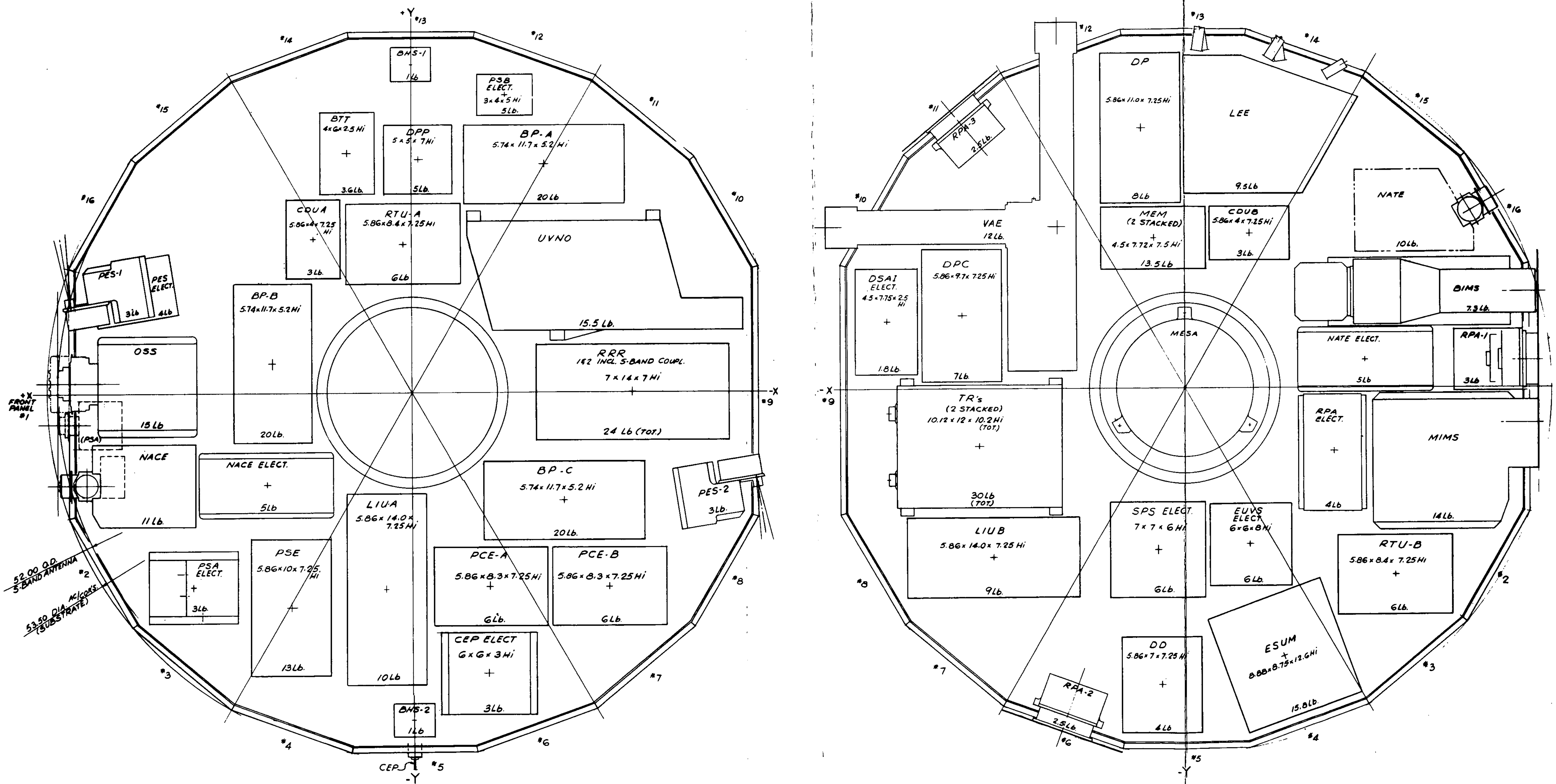


Figure 5. Experiment Layout

1.5.2 Structure

1.5.2.1 Internal Structure

The spacecraft internal structure consists of two 52.5 inch diameter baseplates bolted to a central column. The baseplate mounting surfaces are spaced 17.75 inches apart. Six shear panels tied to and emanating radially from the central column are also tied to the two baseplates. The baseplates are used to mount the electronic equipment and also provide support points for the propellant tanks. The separation ring which interfaces with the booster is bolted to one end of the center column. On the other end of the column, a conical adapter is bolted on and provides the momentum wheel assembly interface.

1.5.2.2 Solar Array Structure

The solar array structure consists of two 16-sided polyhedrons 53.5 inches in diameter at the corner points and 22.5 inches long. One end of each polyhedron is capped with a flat panel. The array structure is composed of individual aluminum honeycomb sandwich panels bonded together to form the composite structure. The array structures are tied to the internal structure by bolting them to the periphery of each baseplate; also the end panel of the lower array structure (+Z side) is bolted to the flange of the center column.

1.5.2.3 Booster Interface

The spacecraft mounts to the booster through an ordnance separable marman clamp, utilizing the standard Delta 18-inch diameter attach fitting configuration.

1.6 Thermal Design

1.6.1 General

The spacecraft thermal control elements comprise rotating type-louver active controllers, propulsion subsystem heaters, baseplate heaters and appropriate insulation and thermal finishes. Temperatures will be within the limits shown in Table 5 for the operating sun-angle range (0° to 90°) and duty cycles.

1.6.2 Active Thermal Controller

The active controller consists of 4 independent louver assemblies mounted between the -Z skin of the spacecraft and the momentum wheel. It provides a means of varying the effective emittance of the -Z surface. This surface controls the overall heat balance of the spacecraft. In order to meet envelope

TABLE 5. THERMAL DESIGN LIMITS

Item	Orbital (°C) Temperature Limits
Batteries	0 to 25
Shunt Limiter	
transistors	-55 to 115
resistors	-90 to 165
Propulsion Subsystem	5 to 30
Tape Recorder	5 to +35
Momentum Wheel Assy.	0 to 20
Experiments	0 to 35
Components	0 to 35
Solar Array	-110 to +150

constraints, a rotary shutter type design approach has been selected. Rotation is achieved by applying heat to a bimettalic spring actuator which supports each louver assembly at its center. The temperature sensing elements which control the current applied to the actuators are themistor type detectors mounted to the upper baseplate, one element being associated with each assembly. Thus, the assemblies are independent affording a measure of redundancy and the capability to compensate to some degree for dissipation variation across the upper baseplate. The controllers are designed to operate full-open to full-closed over a temperature differential of less than 2°C.

1.6.3 OAPS Heaters

The propulsion subsystem contains resistance type heaters mounted to the tanks, thruster, and plumbing as necessary to maintain propellant temperatures above 5°C. The heaters are controlled by redundant sensors* with override by ground command.

*Similar in design to the ATC sensing elements.

1.6.4 Baseplate Heaters

The baseplates contain resistance type heaters which will maintain the spacecraft above allowable minimum operating temperature during periods of low experiment duty cycle. These heaters are also controlled by redundant sensors with override by ground command.

1.6.5 Array Thermal Design

The solar array substrate consists of aluminum honeycomb sandwich with 0.005 inch inner and outer aluminum skins. The core is 3/8 inch on the spacecraft sides, and 1/2 inch on the spacecraft top and bottom panels. The outer skin of the +Z end is conductively coupled to the side array. The thermal mass of the substrate and array is used to maintain temperature transients within limits during low perigee passage.

2.0 SUBSYSTEMS

2.1 Attitude Control Subsystem

2.1.1 Pitch Control Subsystem

The Pitch Control Subsystem provides control about the spacecraft pitch axis in both the spinning and despun modes. The Pitch Control Subsystem consists of the Momentum Wheel Assembly and the Pitch Control Electronics.

The Momentum Wheel Assembly (MWA) contains redundant DC brushless type motors, a magnetic encoder, flywheel and mirror assembly, and two horizon sensors. The MWA maintains spacecraft momentum at a nominal 1200 in-lb-sec. The operating modes consist of a despun mode wherein the wheel spins at 360 RPM ±10% with a corresponding body rate of 1 RPO, (revolution per orbit) and a controlled spinning mode wherein the body rate varies between 0.5 RPM and 10 RPM (in 0.5 RPM steps), with a corresponding wheel rate as low as 40 RPM. The complete range of spin rates is obtained by direct momentum transfer between the wheel and the body. System momentum is maintained via the momentum torquer. Wheel speed control is maintained through speed control circuitry located within the pitch control electronics. In despun mode, any one of 360 different pitch orientations can be selected by ground command.

The earth time signals from two sensors housed within the MWA are digitized aboard the spacecraft and transmitted to the ground as part of the telemetry data. Roll/yaw error magnitude is determined from the difference in the earth transition times, and orbital phase from the anomaly angle. An 'auto-roll' control feature is also provided whereby roll attitude at apogee is computed on board the spacecraft. If roll error exceeds a preset threshold, a magnetic correction torquing program is automatically initiated.

2.1.2 Magnetic Torquing

The control of roll/yaw attitude of the AE spacecraft is achieved by controlled interaction of internally generated magnetic dipoles with the earth's magnetic field. By selecting the proper orbital phasing of spacecraft magnetic dipole polarity switching point (Figure 6) and utilization of time modulation of torquer current (Figure 7) precise orbit average precession of the spacecraft momentum axis is obtained.

Momentum magnitude is similarly adjusted by control of spacecraft magnetic dipole moments which produce torques about the spacecraft spin axis. Momentum magnitude adjustment is obtained in the spinning mode using a magnetometer providing commutation signals by sensing the local earth's field. An alternative approach utilizing the body mounted horizon sensors to provide commutation is currently under study.

2.1.2.1 Roll/Yaw Torquing

The spacecraft contains redundant electromagnet attitude control torquers to maintain the spacecraft momentum axis along the orbit normal. Each electromagnet torquer generates a magnetic moment of approximately 98 ATM^2 (polar orbit) or 160 ATM^2 (near equatorial orbit) and provides a torquing capability of 4.2° per orbit in a polar orbit and 2.5° per orbit in a near-equatorial orbit.

2.1.3.2 Momentum Control

The spacecraft contains redundant electromagnet Spin Control Torquers which are commutated while the spacecraft is spinning. Each electromagnet torquer generates a magnetic moment of 13 ATM^2 (polar orbit) or 47 ATM^2 (near-equatorial orbit) and provides the capability of changing the spacecraft momentum by 1% per orbit in both polar and near-equatorial orbits.

2.1.3 Nutation Damping

The spacecraft contains a silicone fluid filled loop oriented in the YZ plane to provide nutation damping. The damper time constant is 1/3 of the orbit period in the despun mode. The fluid will be contained in two loops to provide redundancy.

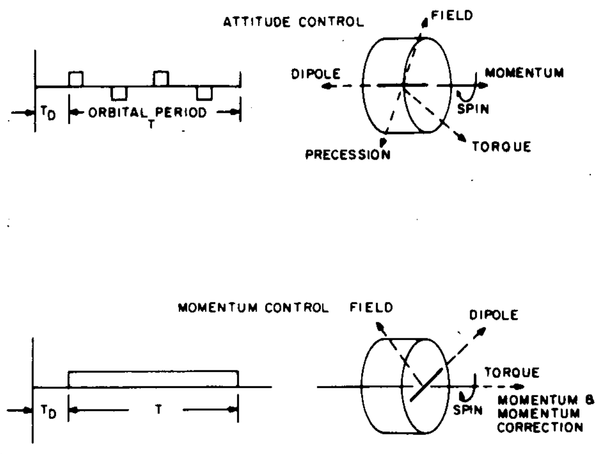


Figure 6. High Inclination Orbit Torquing

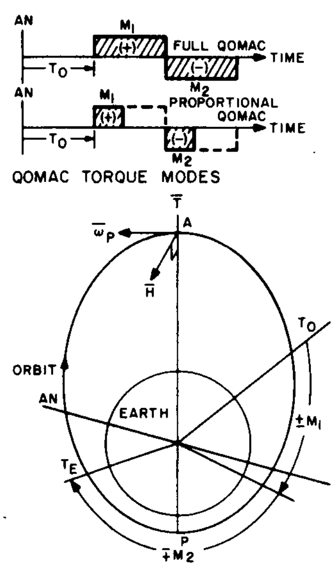


Figure 7. Magnetic Torquing (QOMAC) For Attitude Correction In High Inclination Orbit

2.1.4 Solar Aspect Indication

Solar attitude and spin rate data will be provided by three digital solar aspect indicators equally spaced around the spacecraft cylinder. The sensors provide a coded digital indication of solar aspect angle. The spacecraft also contains a solar gate sensor with a 128° sensitivity plane parallel to the X-Y plane to provide a solar-pulse rotational reference for operations in the spinning mode.

2.1.5 Nadir Indication

Three additional horizon sensors are mounted to the body of the spacecraft, two canted at 70° to the spacecraft -Z axis, one canted at 30° to this axis. From the outputs of each 70° sensor nadir pulse references are derived to provide experiments, in the spinning mode, with a real-time indication of pitch orientation with respect to the local vertical. The desired phase of the pulse delivered to any individual experiment is selectable by ground command.

The 3 body sensors also provide commutation for roll/yaw torquing in near-equatorial orbits, and a back-up technique for attitude determination and OAPS thruster control.

2.2 Orbit Adjust Propulsion Subsystem (OAPS)

2.2.1 General

The OAPS consists of a monopropellant hydrazine/Shell 405 blowdown system. Propellant is stored in positive expulsion tanks symmetrically located with respect to the spacecraft centroid. Redundant thrusters are used to provide a range of thrust varying from approximately 4 lbs to 1 lb dependent upon system blowdown pressure. A third thruster is included on the AE-E mission to provide a means of accomplishing rapid spin axis precession. Total impulse capability of the system is approximately 72,000 lbs sec.

The OAPS accessories include fuel and pressurant fill and drain valves, fuel filtering, tank outlet valves, squib valves, pressure and temperature transducers and associated plumbing. The OAPS is divided into two half-systems of three tanks and one thruster per half-system to provide security against single point failures. These half systems are crossed-coupled via normally open and normally closed squib actuation valves.

2.2.2 Propellant Tanks

Propellant storage is accomplished by 6 Arde' cono-spherical tanks with a total capacity of 370 lbs of fuel. Each tank contains a metal reversing diaphragm for positive fuel expulsion and center of mass management.

2.2.3 Thruster

The thrusters are manufactured by TRW Inc. They are capable of providing multiple starts in either pulsed or steady-state operation over a 6:1 inlet pressure ratio.

2.2.4 Pressurant

The pressurant gas is nitrogen. Pressurant at 600 psi is contained in the ullage space within the propellant tanks and directly pressurizes the propellant throughout the mission.

2.2.5 OAPS Control

Propellant flow is controlled by solenoid valves normally closed (power off). Valves are located at the outlet of each tank. An additional propellant valve is located on each thruster assembly. The command distribution unit contains the power switching for the solenoid valves, squib valves, and the thruster valves. The CDU responds to firing commands from the programmer which will use accelerometer or timer signals for cutoff of thruster operation. In normal operation propellant will be withdrawn from one tank at a time.

2.3 Power Subsystem

2.3.1 Solar Array

The solar array provides a negative-polarity power bus to the spacecraft. Solar cells are mounted on the +Z side of the spacecraft and on the rectangular panels comprising the sides of the cylindrical hat. 4 x 4 cm N-on-P solar cells are used with a number of cells connected in series to be approximately 80 - 84 on the rectangular panels and approximately 117 - 126 on the +Z side of the spacecraft. Blocking diodes are used to protect against the reverse current flow into the cells.

2.3.2 Batteries

Three nickel-cadmium batteries are used to implement the energy storage system. The nominal storage capacity of each battery is six ampere-hours.

2.3.3 Power Supply Electronics (PSE)

The PSE houses in a single container all power supply electronics, except the dissipative components of the shunt limiter. The PSE thus includes: three battery charge controllers, redundant PWM voltage regulator, undervoltage and overvoltage cutoff circuits, fuses, shunt limiter control, and power supply telemetry sensors, and interface circuits. Each battery is charged through a separate charge controller. Each controller circuit monitors signals from third electrode cells within the battery pack which indicate battery state of charge by measurement of gas pressure within the battery cells. Upon reaching a preset pressure level, the charge mechanism will trip the charge to a trickle level for that particular battery. The trip level is varied with temperature to compensate for variation in peak charge with temperature. A Voltage-Temperature tapered charge controller is also included in the system as an optional means of battery charge control. Peak current limit of nominally 1.5 amperes per battery is maintained. Ground command capability to reduce the limit to a lower value (trickle charge) is provided. The PSE contains two pulse-width-modulated voltage regulators with the active regulator selectable by ground command. In the event of an out-of-voltage condition the active regulator is shut-down. The regulated bus provides -24.50 volts ±2% to the housekeeping and experiment loads. An undervoltage circuit is also used to protect against battery overdischarge. Power supply voltage is monitored and, if a lower threshold voltage is detected, all non-essential loads are removed. A number of fuses are provided to remove loads which may exhibit severe, sustained overloads. Only redundant black boxes and non-essential loads are fused. Unregulated bus (solar array) voltage is continuously monitored and, upon reaching a predetermined maximum value, a proportional control signal is delivered to the dissipative components of the shunt limiter. By this means unregulated voltage is restricted to -38.5 volts. Special-purpose circuits are used to receive the output of a number of voltage, current, and temperature sensors, and deliver it to the redundant PCM Assembly in the required form. An ampere hour meter is included in the PSE to monitor state of change of the batteries.

2.3.4 Shunt Limiter

The dissipative components of the shunt limiter are capable of dissipating the maximum solar array power output predicted for the beginning-of-life under the most favorable sun angle condition. They are located on the inside of the bottom array, on the +Z side of the spacecraft.

2.4 Communications Subsystem

2.4.1 VHF Antenna

An omnidirectional VHF antenna is provided for transmission of VHF real time telemetry and beacon. It consists of four dipole elements mounted on the -Z end of the spacecraft array hat, and a coupling network connecting the elements to the output ports.

2.4.2 Dual Mode Beacon/Telemetry Transmitter

Redundant dual mode VHF beacon/telemetry transmitters are used for tracking and real time data readout. For tracking purposes, an unmodulated 0.25 watt carrier is transmitted continuously. Upon command, the power output is increased to 1 watt for transmission of real time data at 16,384 bits/second. A latching relay is used to select the active transmitter and the desired mode of the transmitter. The dual mode transmitter has the following characteristics:

Frequency	136 MHz (exact frequency for AE to be assigned by NASA; identical for both units)
Transmitter power	0.25 watt RF, minimum tracking, 1.0 watt RF, minimum telemetry readout
Modulation	PCM/PM
Modulation Bandwidth	72 kHz
RT data rate	16,384 bps

2.4.3 S-Band Transponder and Coupling Assembly

Redundant S-Band Transponders perform the simultaneous functions of command reception, coherent turnaround ranging and transmission of real time and playback telemetry. They are associated with a frequency diplexer, a 3 dB hybrid coupler and a switchable circulator which together comprise the S-Band Coupling Assembly.

The receiver portion of the transponder is a high sensitivity superheterodyne employing a phase lock carrier tracking loop, a separate wideband (modulation) phase detector, IF limiting and a coherent AGC detector. The transmitter is a solid state unit incorporating a linear phase modulator and having the capability of generating an output RF carrier coherently related to the received frequency or, on command, derived from a local frequency source. The transmitter has provisions for automatic turn-on upon receipt of an uplink RF carrier.

Major characteristics of the transponder are as follows:

Receive Frequency 2108.25 MHz

Transmit Frequency 2289.50 MHz

RF Output Power 5 watts

Modulation Bandwidth 4 kHz to 1.4 MHz

DC Power Input 40 watts.

Modulation:

Telemetry	PCM/PSK/PM
Command	PCM/FM/PM
PRN Ranging	PM on carrier
GRARR Ranging	CW/PM
Data Rate	16,384 bps (RT); 131,072 bps (PB)

2.4.4 Dual Pre-Modulation Processor (PMP)

The function of the PMP is to process and combine RT & PB telemetry data prior to phase modulation of the RF carrier. It phase modulates a 1024 kHz subcarrier with the 16384 bps biphase bit stream, phase modulates a 768 kHz subcarrier with the 131072 bps biphase bit stream, combines them and furnishes the composite signal as an output to the S-Band transponders.

2.4.5 S-Band Antenna

An S-band belt-antenna is provided for command reception, ranging and S-Band telemetry transmission. It is a resonant cavity slot-type design consisting of six segments mounted to the equator of the spacecraft. The feeds are appropriately phase related to obtain an omni-directional pattern with 6 dB maximum ripple.

2.5 Data Handling

2.5.1 PCM Assembly

The PCM assembly converts analog, discrete, and digital science and status telemetry inputs into a uniform serial bit stream, delivered at a rate of 16,384 bits/sec to either of the two GFE tape recorders, to either of the two redundant VHF transmitters, or to either of the two S-band transponders. It provides four sets of 64 or 128 channel subcommutation, 128-channel main frame commutation, A/D conversion, and encoding. Timing is derived from a redundant spacecraft crystal oscillator timing source contained in a logic interface unit.

The assembly consists of 3 units, two remote telemetry modules (RTM's) one per baseplate, and a central dual-PCM controller (PCMC). Each RTM, which is piecewise redundant, collects and multiplexes analog, digital and discrete data, from equipments mounted locally, under the control of the PCMC. The PCMC A to D converts the analog data and assemblies all telemetry into a single serial bit stream. A read/write memory module is included in the controller to permit reprogramming of the main-frame format.

2.5.2 Tape Recorder (GFE)

Two tape recorders are used to store science and status data. Either recorder (or both for sequential operation) can be selected by ground command to enter either the playback or the record mode. After the record or playback cycle is completed, the recorder automatically re-sets to standby.

Each recorder has a total bit storage capacity of approximately 120×10^6 bits, and has the following characteristics:

Playback-Record Speed-Up Ratio:	8:1
Record Bit Rate:	16,384 bits/sec
Error Rate:	1 in 10^5 bits
DC Power:	4.5 watts (Record)
	9.2 watts (Playback)

2.6 Command and Control

The capability to deliver both real time and stored commands at either relay-driving or logic level are provided in the AE spacecraft. Random access memories are included in the system permitting maximum flexibility in terms of experiment and spacecraft support equipment remote programming.

Commands are of four general types:

- Power Commands (turn on/turn off commands)
- Major mode (ie pulse) commands at high-voltage level (-24 volt or greater)
- Major mode commands at logic level
- Minor mode commands.

The last-named, which consist of up to 32 bits of serial data, can be supplied to any user for subsequent decoding by that user as required. In this manner the effective total number of 'commands' obtainable is considerably increased.

Principal elements of the command and control equipment are as follows.

2.6.1 Dual Demod/Decoder

The dual demod/decoder (DD) is compatible with the Apollo command standards which are to become the basis for the consolidated STADAN/MSFN network. The DD demodulates the 70 kHz command subcarrier output from the S-Band receivers, extracts command information and the transmitted clock signal, checks overall code validity and transmits data to either of two 32 kbit memories or to the logic interface units for real time execution. Each DD is associated uniquely with one receiver.

The command word is 64 bits long, transmitted at a data rate of 1024 bps. 'Sync', preceding a command stream, consists of at least 13 zeros followed by a 'one'. The command word itself contains a 7 bit spacecraft address, 2 bits to identify the DD to be utilized, a 9 bit op-code, a 32 bit minor mode command if used (or a 32 bit spacer) and, a 7 bit check code. Other bits are used as required for internal functions. Stored Commands are placed in memory in the form of minor mode commands and in this case contain an op-code, time-tag information and minor mode information if required. The 9 bit op-code, which represents a major mode command, either from memory or real time, is transmitted serially to the logic interface units where it is code translated to a 2 out of 32 format.

2.6.2 Logic Interface Units (LIU's) and Command Interface Units (CDU's)

These units, which are piecewise redundant one each per baseplate provide the 'fan-out' for commands to individual equipments; the former for logic level commands, the latter for high-voltage level commands. The LIU translates the 9 bit op-code to 2 out of 32 matrix. The matrix is operated upon in the LIU's to derive logic level major mode commands directly and is also applied to the CDU's for generation and distribution of the high-voltage (relay-driving type) commands. Each LIU also distributes minor mode data, furnishes word-enables for telemetry readout, supplies general spacecraft timing information and houses a read-only-memory module for SPS control.

2.6.3 Dual Programmer

The dual programmer (DP) provides control of memory operations and also performs certain special functions which are not adaptable to time-tag memory control. Among these are included nadir pulse generation, OAPS ΔV control, auto-roll calculations and other sundry operations related to attitude control.

2.7 Solar Pointing Subsystem

The solar pointing subsystem (SPS) consisting of a two-axis gimballed platform and associated electronics maintains sun-orientation of the EUVS experiment. The integrated assembly is housed in the spacecraft adapter section of the spacecraft. Gimbal drive servos, under the control of sun sensors, mounted on the experiment, can maintain pointing to within one minute of arc of the center of the sun, or, alternatively, provide offset pointing and raster scan modes of operation upon demand. Up to 40 slip rings are provided in the azimuth drive assembly to accommodate experiment electrical interfaces. This subsystem was originally treated as part of the experiment complement.

2.8 Engineering Measurements

Two pressure sensors are supplied as GFE to measure the dynamic pressure in the vicinity of perigee. One is a capacitance manometer type, the other an ion gauge.

2.9 Experiments

The payload experiments for each of the three Atmosphere Explorer missions (AE-C, D, & E) are listed in Table 6. Table 7 lists the principal physical parameters of the experiments.

TABLE 6. EXPERIMENT PAYLOADS FOR AE MISSIONS

Investigator	Experiments	AE-C	AE-D	AE-E
L. Brace	Cylindrical Electrostatic Probe Experiment (CEPE)	x	x	x
K. Champion	Atmospheric Density Accelerometer (MESA)	x	x	x
J. Doering	Photoelectron Spectrometer (PES)	x	x	x
W. Hanson	Retarding Potential Analyzer (RPA)	x	x	x
D. Heath	Extreme Solar Ultraviolet Monitor (ESUM)	x	x	x
H. Hinteregger	Extreme Ultraviolet Spectro-photometer (EUVS)	x	x	x
A. Nier	Open-Source Neutral Mass Spectrometer (OSS)	x	x	x
D. Pelz	Neutral Atmosphere Composition Experiment (NACE)	x	x	x
N. Spencer	Neutral Atmosphere Temperature Experiment (NATE)	x	x	x
J. Hoffman	Magnetic Ion Mass Spectrometer (MIMS)	x	x	
H. Brinton	Bennet Ion Mass Spectrometer (BIMS)	x		x
C. Barth	Ultraviolet Nitric Oxide Spectrometer (UVNO)	x	x	
R. Hoffman	Low Energy Electron Experiment (LEE)	x	x	
P. Hays	Visual Airglow Experiment (VAE)	x	x	x
J. Armstrong*	Magnetometer		x	x

*This experiment was added at the time of preparation of this report. Interface data is not included herein.

TABLE 7. PRINCIPAL EXPERIMENT PARAMETER

Expt	Weight (lbs)	Power (watts)	No. of Power and Major Mode Commands
RPA	11.0	6.0	13
MIMS	10.0	7.0	3
BIMS	7.6	2.0	5
PES	10.0	2.5	10
MESA	16.0	19.5 (unreg)	24
OSS	15.0	8.0	10
NACE	16.0	16.0	10
NATE	15.2	16.0	8
ESUM	18.1	2.25	17
EUVS	24.0	12	32
UVNO	15.5	8.2	4
VAE	12.0	7	13
CEP	4.0	2	21
LEE	9.5	2.5	7
Totals	184	95 reg +19.5 unreg	177